PART 6 – PRIME POWER SYSTEMS

Section

PART 6 – PRIME POWER SYSTEMS

INTRODUCTION

Three types of prime power systems are discussed: Solar power systems, nuclear power systems, and chemical power systems.

Mission constrains of duration, power requirement, environment, and goals play an important role in the selection of the prime power source or sources. The three types of power sources used in space missions to date are documented in this part. Some implementations of these types of sources have been flown but not all, where applicable this is noted.

Solar Power Systems

The most useful means of converting solar power into electric power is by means of solar cells. Other means include solar thermoelectric systems, solar thermionic systems, and solar dynamic systems.

Nuclear Power Systems

The heat of nuclear fission is used to generate electric power. This may be done passively as in a radio isotope source or activity as in reactor dynamic power systems. These and different methods of converting the heat energy into electric energy are discussed.

Chemical Power Systems

Chemical power systems are in the form of batteries and fuel cells. The lifetime, cycling capability and capacity of these sources are given.

SUMMARY

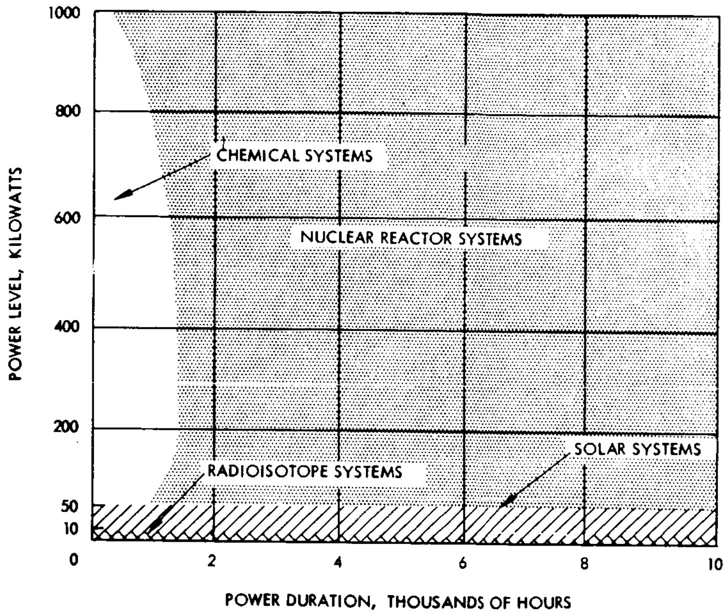
The type of power source used depends upon the mission, its duration and the power requirements.

The selection of a spacecraft power system for a specific mission depends on the power requirements of the mission, the mission duration, and the environment in which the system must operate. The effect of these constraints on the selection of power system of a few watts to kilowatts capacity is discussed in this Part.

Spacecraft power systems may be classified into three general categories according to the initial energy source as solar, nuclear, or chemical. The weight of solar and nuclear systems is generally not a strong function of mission duration, whereas the weight of the chemical system is decidedly so. Typical power system selections based on a solar distance of 1 AU* are shown in the figure[1] as a function of power level and mission duration. Nearly every power system will include some provision for energy storage to provide for peak power demands and, in the case of solar systems, to provide continued power during periods of solar eclipse. The extent and type of energy storage required depends critically on the exact mission power history and, in the case of solar systems, on the solar illumination history.

*1AU (astronomical unit) ≈ 149.6×10^6 km.

[1] 1967 NASA Authorization, Part 4, United States Government Printing Office, Washington, D.C., 1966.



Power System Range of Application at Near-Earth (1 AU) Solar Distance

PRIME POWER SYSTEMS

Solar Power Systems

SOLAR VOLTAIC SYSTEMS

Solar cells are very useful as power sources in space. Constants are given which relate weight, power, cost and area of solar cells.

Basically there are two types of solar systems: photovoltaic and solar thermal systems. The first group will be discussed in this topic and the second group in a later topic. The latter group includes solar thermoelectric and solar thermionic systems as well as solar dynamic systems of various types.

Solar photovoltaic (solar cell) systems are presently by far the most appealing solar power system. At present they alone have proven reliability. They are considerably more efficient (hence lighter and more compact) than solar themoelectric systems. By comparison with solar thermionic and solar dynamic systems, they are relatively insensitive to angular orientation with respect to the incident solar illumination. Photovoltaic array power output is directly proportional to surface area (hence weight) to powers of many kilowatts.

Over the range of solar illumination for which photovoltaic arrays are useful, cell efficiency is a function only of array temperature, i. e., power output is directly proportional to illumination intensity. Hence, for constant array temperature, the specific weight increases as the square of the solar distance. Conversely, the specific power (watts/kg) decreases as the square of the solar distance. Array specific power at constant illumination decreases as cell temperature increases. This is because solar cell efficiency is an inverse function of temperature as seen in Figure A. As a result, photovoltaic array power passes through a maximum as solar distance is reduced to about 0.5 AU and then is sharply reduced (Figures B and C).[1] Solar array weight and area as a function of unregulated output power are shown in the table for various solar distances based on expected capability in the near future. Solar photovoltaic power system cost constants are also given in the table. These estimates include the solar cells, supporting structure, mechanisms, and wiring, but not power conversion and distribution equipment.

At intermediate power levels, a choice must be made between an oriented or a non-oriented photovoltaic array and between various geometric array configurations. The choice depends on 1) the trade-off between decreased array weight, size, and cost and increased system complexity and development cost with orientation, 2) limitations on array moment of inertia imposed by the vehicle attitude control system, and 3) packaging limitations.

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[1] Gross, Sidney, "Discussion of Power Systems for Solar Probes — Solar Photovoltaic Concepts," Proceedings of the Intersociety Energy Conversion Engineering Conference, Los Angeles, California, September 26-28, 1966.

Table A. Solar Cell Power, Weight, and Cost Parameters

Position of Solar Cells	Power Output Watts/cm^2	Weight Kilograms/watt	Cost Dollars/watt
Mercury (.39AU)	1.87×10^{-2}	0.0186	43
Venus (.72AU)	2.10×10^{-2}	0.01775	38.3
Earth (1.0AU)	1.52×10^{-2}	0.0227	53
Mars (1.52AU)	0.72×10^{-2}	0.0481	112
Jupiter (5.2AU)	0.06×10^{-2}	0.575	1350

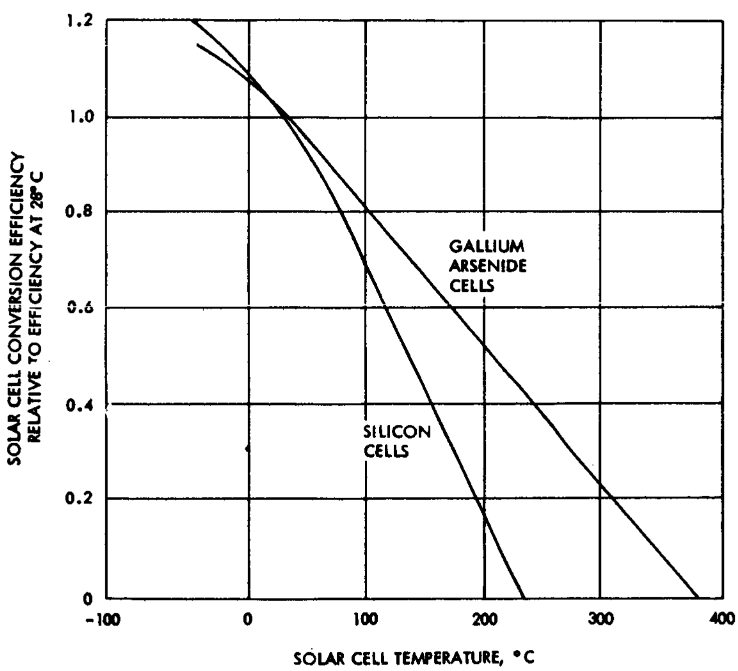

Figure A. Temperature Dependency of Solar Cell Efficiency

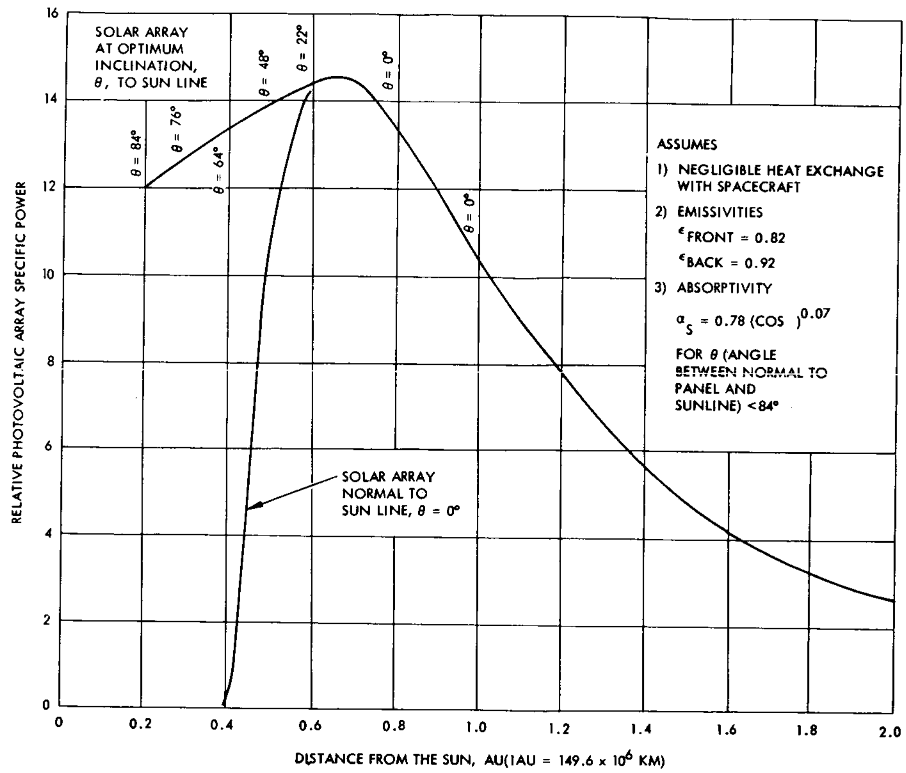

Figure B. Photovolaic Array Specific Power Versus Solar Distance — Silicon Cells

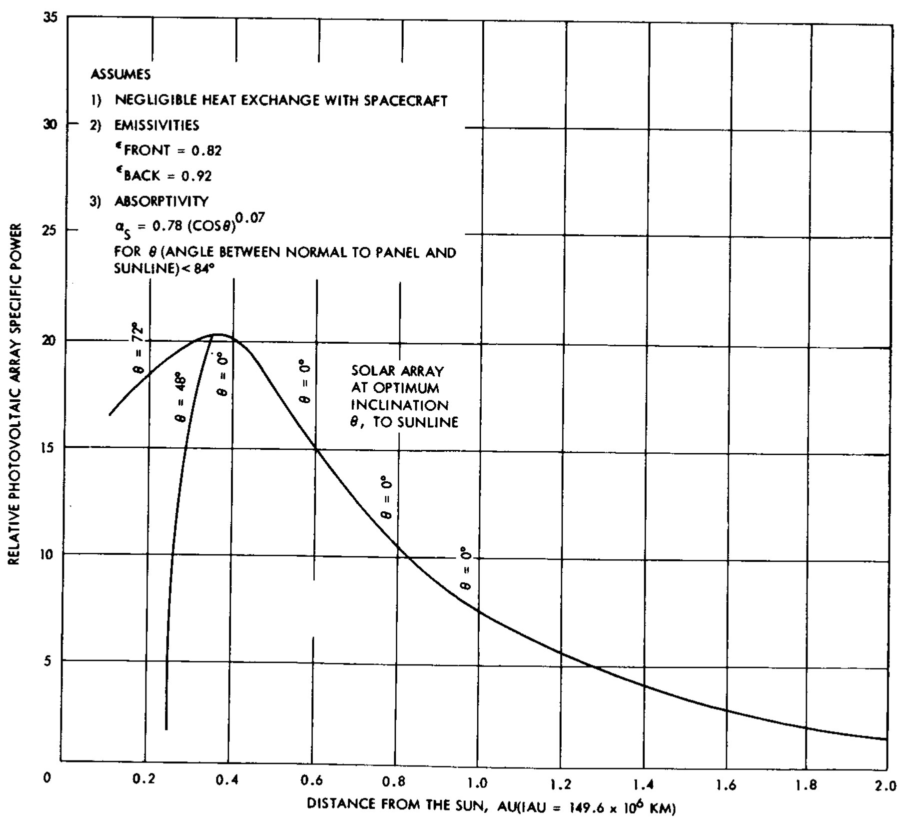

Figure C. Photovoltaic Array Specific Power Versus Solar Distance — Gallium Arsenide Cells

SOLAR CELL DEGRADATION IN A SPACE ENVIRONMENT

The amount of solar cell degradation is estimated, based upon a March 1969 launch to Mars.

Solar cell performance is degraded by the particle radiation environment encountered in space. The conventional technique for protection of solar cells against the degrading effects of particle radiation is the use of transparent cell covers as shielding. The degree of protection from radiation damage afforded by such methods depends of course on the thickness of the covers as well as upon the protective properties of the material used. A supplementary technique consists of overdesigning the array to an extent that it will maintain an acceptable power output for the duration of the mission after being degraded to the degree predicted on the basis of the known or anticipated radiation environment. For a given radiation environment and power requirement, there is an optimum compromise for minimum weight between increasing cell cover thickness to reduce degradation and increasing the size array so that it will continue to deliver the required power after being degraded. A study of solar array degradation as a function of time during Earth-Mars transit due to solar flare activity has been made for 30- and 45-mil quartz covers on the basis of an assumed March 1969 launch date. The principal cause of array degradation by solar flares is proton radiation. Thirty- and 45-mil quartz covers are impervious to protons having energies less than 10 mev and 15 mev, respectively. Tables A and B list approximate percent of initial solar array power capability remaining in successive months following the March 1969 launch for 30- and 45-mil quartz covers.

Table A. Solar Array Degradation in Earth-Mars Transit by Solar Flare Activity Using n on p Silicon Cells with 30-Mil Quartz Covers

Month	Total Proton Flux (Protons/Cm^2 having E > 10 mev)		Percent of Original Power Capability Remaining	
	Maximum	Minimum	Maximum	Minimum
1969				
March (Launch)	0	0	100	100
April	5×10^{9}	2.8×10^{9}	95	93
May	1.4×10^{10}	5×10^{9}	93	90
June	1.8×10^{10}	8×10^{9}	91.3	88
July	3×10^{10}	1.4×10^{10}	90	86.6
August	4×10^{10}	1.6×10^{10}	89	85.2
September	5×10^{10}	1.8×10^{10}	88	84.2
October	5.5×10^{10}	2.5×10^{10}	87.2	83.2
November	6.0×10^{10}	3×10^{10}	86.8	82.6
December	6.3×10^{10}	3.4×10^{10}	86.0	82.0
1970				
January	7×10^{10}	4×10^{10}	85.7	81.0
February	7.5×10^{10}	4.4×10^{10}	85.0	80.6
March (Intercept)	8×10^{10}	4.8×10^{10}	84.6	80.0

Table B. Solar Array Degradation in Earth-Mars Transit by Solar Flare Activity Using n on p Silicon Cells with 45-Mil Quartz Covers

Month	Total Proton Flux (Protons/Cm^2 having E > 10 mev)		Percent of Original Power Capability Remaining	
	Maximum	Minimum	Maximum	Minimum
1969				
March (Launch)	0	0	100	100
April	2.5×10^9	1.4×10^9	96	95
May	7×10^9	2.5×10^9	95	92
June	9×10^9	4×10^9	94	91
July	1.5×10^{10}	7×10^9	92	89
August	2×10^{10}	8×10^9	91.5	87.5
September	2.5×10^{10}	9×10^9	91.0	86.5
October	2.75×10^{10}	1.25×10^{10}	90.0	86
November	3×10^{10}	1.5×10^{10}	89.0	85.8
December	3.15×10^{10}	1.7×10^{10}	88.5	85.6
1970				
January	3.5×10^{10}	2×10^{10}	87.5	85.4
February	3.75×10^{10}	2.2×10^{10}	87.3	85.3
March (Intercept)	4×10^{10}	2.4×10^{10}	87.0	85.1

SOLAR THERMAL SYSTEMS

Solar thermoelectric, thermionic, and dynamic systems are described. They may find greatest use in a high solar flux near the sun.

Solar Thermoelectric Systems. Solar thermoelectric systems consist of thermoelectic elements heated by solar illumination either directly or using concentrators. In either case, the low efficiency of the thermoelectric conversion process (5 percent) leads to specific weights upward of 91 kg/kw[1] (200 lb/kw) or 10.1 watts/kg (5 watts/lb) at 1 AU solar distance. They may, however, be competitive with photovoltaic systems for solar probes where solar cell efficiency is severely degraded by high temperature. Estimated solar thermoelectric power system weight and area interdependancies upon power are given in the Table for 1 AU and 0.3 AU solar distances.[2]

Solar Thermionic Systems. Solar thermionic systems use concentrating mirrors to focus solar energy on thermionic converters. Although they are not competitive with photovoltaic systems on a weight basis, considerable developmental effort has been expended in the hope that they will be superior in high radiation environments or high operating temperatures (such as might be encountered on solar probe missions). Orientation accuracies required for the large collectors (5 minutes of arc) and the problems associated with their deployment and possible degradation by the space environment pose severe problems. Solar thermionic systems are still in too early a stage of development for accurate evaluation. Estimated solar thermionic power system weight and area interdependancies on power are also noted in the Table for solar distance regimes of 0.1 to 0.3 AU and 0.35 to 0.7 AU.[2]

Solar Dynamic Systems. Solar dynamic systems are characterized by the use of a heat engine (typically a turbine) to drive an electrical generator. Solar Brayton cycle systems are regarded as promising for high-temperature and high-radiation environments, but share with solar thermionic systems the problems associated with deployment and orientation of large collecting areas in space. They are not sufficiently developed to permit accurate evaluation.

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[1] Rappaport, Paul, "Space Power: The Next Step," Space/Aeronautics, 45, Number 4, 1. 76, September 1965.

[2] Brosens, P. J., "Discussion of Solar Power Systems for Solar Probes — Thermoelectris and Thermionics," Proceedings of the Intersociety Energy Conversion Engineering Conference, Los Angeles, California, September 26-28, 1966.

Power, Weight and Area Interdependacies for Thermoelectric and Thermionic Power Systems

Distance from the Sun, AU	Thermoelectric Power		Thermionic Power	
	Watts/cm^2	Kg/cm^2	Watts/cm^2	Kg/cm^2
1.0	1.66 x w^{-2}	0.09		
0.3	2.54 x 10^{-2}	0.051		
0.7 to 0.35			2.06 x 10^{-3}	0.0373
0.3 to 0.1			6.36 x 10^{-2}	0.0256

PRIME POWER SYSTEMS

Nuclear Power Systems

NUCLEAR POWER SYSTEMS

INTRODUCTION TO NUCLEAR POWER SYSTEMS

The advantages of nuclear power systems include their long life and independence of solar radiation.

Nuclear power systems convert the thermal energy generated by nuclear reactors or isotope decay to electrical energy by the same conversion cycles employed with solar thermal power systems, i.e., turboalternators, thermionics, or thermoelectrics. The primary advantages of nuclear systems over solar systems are the independence from solar illumination and their potentially lower specific weight at high power levels. Their main disadvantage is the nuclear radiation produced and shielding required to protect personnel or radiation-sensitive equipment from it.

Reactor power sources are generally applicable to high power levels (about 10 kw), and may be competitive at much lower levels for deep space missions. This results from the relatively high specific weight for low power designs due to the heavy reactor and shielding. Problems with reactor system reliability and life have been experienced partly because of the necessity to integrate sophisticated nucleonic control systems and high temperature fluid systems.

Reactor problems may be avoided by using radioisotopes as energy sources because their thermal energy output is continuous and predictable. Radioisotope thermoelectric systems can be designed for very low power levels and like reactors can operate for very long times. Although their specific weight is one of the lowest for power levels from 1 to 10kw, radioisotope systems are seldom used where alternative systems can provide competitive performance. One reason for this is the higher cost of radioisotope systems resulting from the limited quantities of isotopes available; another is the complications introduced by radiation produced.

THERMOELECTRIC REACTOR POWER SYSTEMS

The theory of reactor power systems operation is given with data on performance.

Reactors can serve as a heat source for thermoelectric converters. Thermoelectric converters transform heat energy to electrical energy by means of the Seeback effect in a thermoelectric couple. The configuration of a thermoelectric converter element is shown schematically in Figure A. Heat is transferred to the thermoelectric elements P and N (two dissimilar conductors or semiconductors) through the copper block on top and rejected through the two copper blocks at the bottom. As a result of the Seeback effect, a voltage is developed between the bottom ends of the two thermoelectric elements. In practical converters, large numbers of such elements are combined in series as in Figure B and then in parallel to produce usable voltages and currents.

A typical thermoelectric power system is illustrated schematically in Figure C. An actual system configuration is shown in Figure D. Because of the relatively low efficiency of thermoelectric conversion (5 to 8 percent) extensive radiator area is required. A 570-watt reactor thermoelectric power system has been successfully tested in space for 43 days and has operated up to 10,000 hours in a simulated space environment. System parameters, including weight, cost, and volume parameters, for reactor thermoelectric systems of the SNAP 10A type developed by Atomics International are shown in the Table[1] for various power levels.

[1] Glyfe, J.D., and Wimmer, R.E., "Reactor Thermoelectric Power Systems for Unmanned Satellite Applications," Proceedings of the Intersociety Energy Conversion Engineering Conference, Los Angeles, California, September 26-28, 1966.

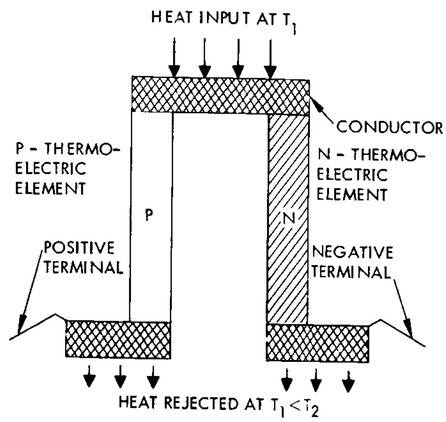

Figure A. Basic Thermoelectric Couple, Schematic Arrangement

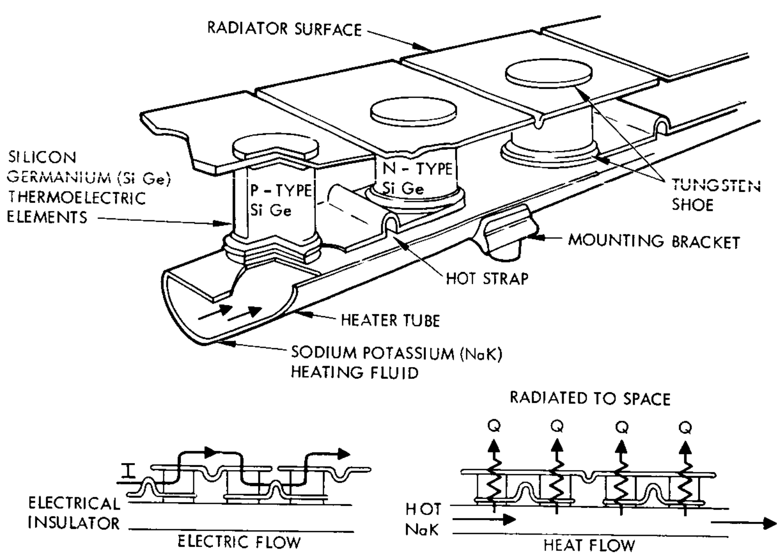

Figure B. Direct Radiating Thermoelectric Module

THERMOELECTRIC REACTOR POWER SYSTEMS

Reactor Thermoelectric System Performance (Atomics International)

Net Power, Electrical (kw)	0.5	1.0	2.0	5.0	10.0	15.0	20.0
Reactor Power, Thermal (kw)	28.4	47.1	84.8	212	424	635	850
Reactor Outlet Temperature (°F)	1300	1300	1300	1300	1300	1300	1300
Design Life, Rated Power (yr)	3	3	3	3	3	3	3
Gross Radiator Area (ft^2)	24	48	98	245	525	865	1150
(cm^2)	2.23×10^4	4.46×10^4	9.1×10^4	22.8×10^4	48.8×10^4	80.4×10^4	107×10^4
Base Diameter (ft)	2.84	3.67	4.84	7.3	10.0	12.7	15.2
(cm)	86.7	111.5	147.5	222.5	305	388	463
Overall Height (ft)	7.67	10.17	13.58	21	31.5	41.5	50.5
(cm)	234	310	414	641	960	1265	1540
Unshielded System Weight (lb)	524	633	852	1740	2885	4215	5850
(kg)	238	288	387	791	1310	1915	2660
Reactor – Payload Separation Distance (ft)	45	52	65	80	100	100	100
(cm)	13.7×10^2	15.9×10^2	19.9×10^2	2.48×10^2	30.5×10^2	30.5×10^2	30.5×10^2
Total Shielded System Weight (lb)	685	829	1076	2385	3670	5060	6830
(kg)	311	377	489	1085	1670	2300	3100
Specific Power (10^3 lb/kw)	1.37	0.83	0.54	0.48	0.37	0.33	0.32
(kg/kw)	623	377	245	218	168	150	145
Specific Cost* (10^3\$/kw)	1600			600		470	450
	1250			400		310	300

*Upper figure is current specific cost; lower figure is potential specific cost with quantity production.

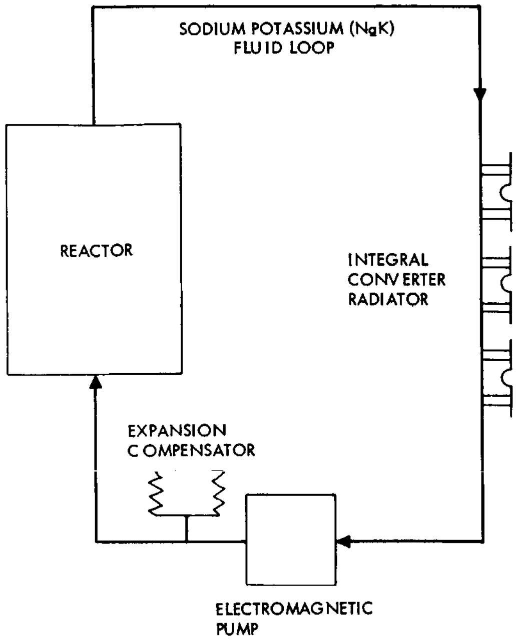

Figure C. Reactor-Thermoelectric System Schematic

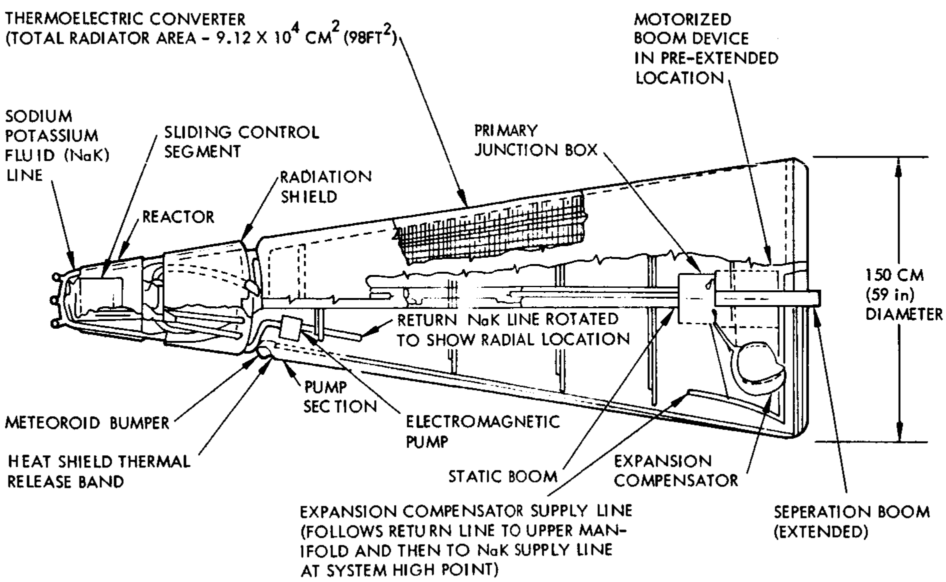

Figure D. Two Electrical Kilowatt Reactor Thermoelectric Power System (Atomics International)

THERMIONIC REACTOR SYSTEMS

Thermionic systems use thermionic diodes heated by the reactor to produce electricity. Temperatures in the order of 3000°C are used.

A thermionic diode converter can be used to convert heat into electricity. The device is illustrated schematically in Figure A. A hot cathode emits electrons which travel across a narrow gap to a relatively cool anode. If the cathode and anode are connected externally, the electrons collected on the anode return to the cathode through the external circuit. Thus an electric current $\underline{i}$ is established. If a load is inserted in the external circuit, a potential difference is developed across it with the signs as indicated. Insofar as the external circuit is concerned, the cathode is the positive terminal and the anode is the negative terminal of the thermionic generator.

Heat is continually supplied to the cathode to compensate for the energy taken by the emitted electrons and loss of heat from the cathode due to radiation, convection, and conduction. As the external surface of the cathode is entirely used for transfer of heat from the heat source to the cathode, this loss of heat from the cathode is mostly transferred to the anode. They may be rejected in order to keep the anode temperature from rising too high.

The thermionic diode is a heat engine: thermal energy is supplied to the cathode at a high temperature T_2 and a portion of it rejected from the anode at a lower temperature T_1. The conversion efficiency is proportional to $(T_2 - T_1)/T_1$; the greater the temperature difference $T_2 - T_1$, the more efficient the diode is. Thus a reactor is theoretically well-suited as a thermionic heat source since it can provide heat at high emitter temperatures with the resultant increased efficiency and decreased specific weight. The major problems associated with thermionic converters concern the very high emitter temperatures that are required to obtain lightweight systems and the resultant reliability problems. Various reactor thermionic system configurations are illustrated in Figure B. Of these, the converter integral with the reactor offers potentially the lowest specific weight and volume, but poses the greatest reliability problems because it operates with the highest emitter temperature. The thermionic converter has higher efficiency (25 to 30 percent at 2000°C emitter temperature) than thermoelectric converters (5 to 8 percent) because of its high operating temperatures. Its higher operating temperatures also permit higher heat rejection temperatures which reduce the radiator area requirements. Reactor thermionic systems have not yet reached the flight hardware phase and are not likely to be adequately developed for flight use before the mid 1970's. It is not possible at this time to estimate accurately costs or volumes of space qualified reactor thermionic systems. Estimated weights relationships of reactor thermionic systems for various emitter operating temperatures are shown in the table.[1]

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[1] 1967 Authorization, Part 4, United States Government Printing Office, Washington, D.C., 1966.

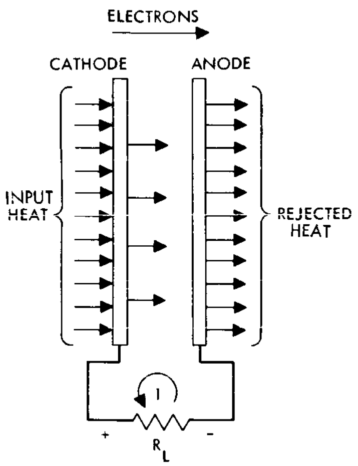

Figure A. A Thermionic Diode Converter of Heat to Electrical Energy

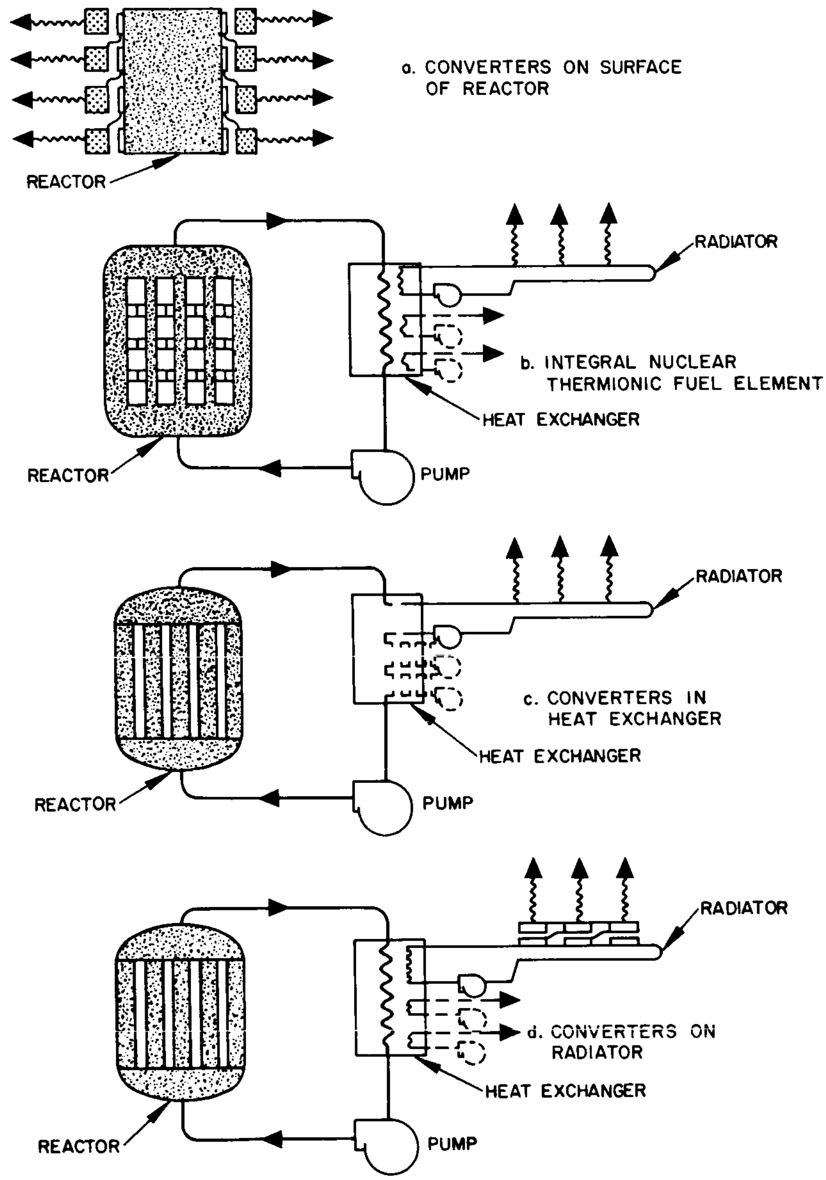

Figure B. Reactor Thermionic Power Systems

Nuclear Thermionic Power/Weight Relationship

Emitter Temperature, °C	Kilograms/Kilowatt	Pounds/Kilowatt
2700	38.6	85
2800	29.5	65
3000	18.2	40

REACTOR DYNAMIC POWER SYSTEMS – BRAYTON CYCLE

A Brayton cycle dynamic reactor using a turbine is described which produces powers in the order of 3 kw.

Dynamic power systems are characterized by the use of a heat engine (reciprocating engine or turbine) to drive an electrical generator. Reactor dynamic power systems appear attractive for high power requirements. Although no complete power system is presently being developed, component development continues on several types. The two heat engine types which are presently being studied are the Brayton cycle and the Rankine cycle. The Brayton cycle is described below and the Rankine cycle is described in the next topic.

The working fluid for the Brayton cycle is an inert gas such as argon or neon. Heat input is at constant pressure from a suitable heat source. The hot gas is expanded through a turbine and the waste heat is rejected in a radiator at a continuously decreasing temperature. The gas is compressed and the cycle repeated.

The principal advantages of the Brayton cycle are

1. The inherent simplicity of a single loop and a single phase working fluid enhances the system reliability.

2. The corrosion free atmosphere provided by the inert gas will allow use of uncoated high temperature refractory alloys without fear of corrosion or oxidation.

3. Similarly, the inert gas system should also be erosion free because there will be no solid or unburned particles in the working fluid.

Its major disadvantages are

1. Since most of the heat of the cycle is not added at the highest temperature or rejected at the lowest temperature of the cycle, the efficiency of the simple Brayton cycle is low. However, this can be largely offset by the proper selection of cycle temperatures and the use of a regenerator.

2. Considerable pumping power is required for the compression process in the Brayton cycle as compared to pumping a liquid in the Rankine cycle.

3. The continuously decreasing temperature in the radiator and the low radiator outlet temperature increase the heat rejection problem. Since the only means of heat rejection is by radiation, a large radiating area is required.

A module of a SNAP 8 reactor, powered 10-kw Brayton-cycle system, is shown schematically in the figure.[1]

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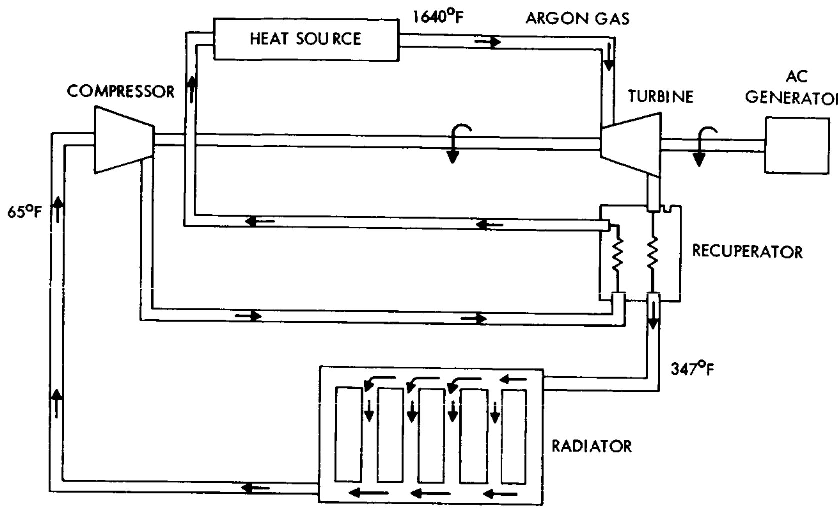
[1] 1967 Authorization, Part 4, United States Government Printing Office, Washington, D.C., 1966.

The module has a design weight of approximately 330 kg (725 pounds) or 33 kg/kw (72.5 pounds/kw) and requires a radiator area of approximately $5.57 \times 10^5 \text{ cm}^2$ (600 ft^2). The measured efficiency of a 3-kw demonstration Brayton-cycle system was 18 percent (electrical output/heat input). No flight tests of a complete system are presently scheduled, and an operational system is not expected until the 1970's.



Brayton Cycle Dynamic Power System

REACTOR DYNAMIC POWER SYSTEMS — RANKINE CYCLE

Rankine cycle power systems have a higher efficiency than Brayton cycle systems. Prototype units have produced 10 kw with an unshielded weight of 2000 pounds.

As mentioned in the previous topic reactor dynamic power systems use a reciprocating engine or a turbine operating from heat produced by a reactor. Heat input to the Rankine cycle is used to vaporize and, if required, superheat the working fluid until the desired conditions are achieved at the turbine inlet. The waste heat at the turbine exhaust is dissipated by radiation to space until the fluid is completely condensed to a liquid. The liquid is then pumped to the boiler where the cycle is repeated. Superheating is generally required to prevent the possibility of any vapor condensing during the expansion process which would cause erosion and a reduction of prime mover efficiency. A wide variety of working fluids can be used, but for space applications, liquid metals and organic fluids are receiving the most attention.

The principal advantages of the Rankine cycle are

1. The efficiency approaches that of the Carnot cycle since most of the heat is added isothermally and most of the waste heat is rejected isothermally.

2. Isothermal rejection of waste heat is desirable from the standpoint of minimizing the radiator area. In addition, the heat rejection temperature can be considerably higher than for the Brayton or Stirline cycles.

3. The Rankine cycle has received the greatest development effort.

The principal disadvantages of the Rankine cycle are

1. The cycle is mechanically comples, two or more loops may be required, particularly in the case of nuclear heat sources.

2. The corrosion and erosion problems associated with metal vapors may adversely affect the system life.

A Mercury-Rankine component development program has been underway for a number of years. The objective has been to develop and qualify components for a basic 3 to 5 kw module. A typical Reactor Mercury Rankine power system based on present components is shown in the figure.[1] The reactor is used to heat sodium-potassium (NaK) fluid in a sealed recirculating loop. The hot sodium-potassium is circulated through a boiler to evaporate and superheat the mercury working fluid. The mercury vapor is then expanded within a turbine which drives a

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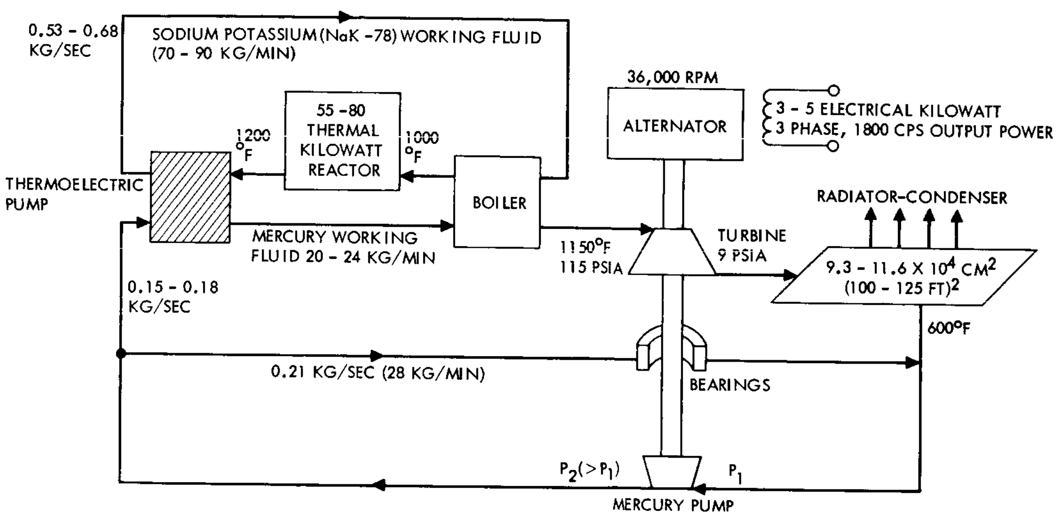
[1] Wallerstedt, R. L., and Owens, J. J., "SNAP Mercury Rankine Program," Intersociety Energy Conversion Engineering Conference, Los Angeles, California, September 26-28, 1966.

permanent magnet alternator and a mercury boiler feed pump. These three components are all mounted on a single rotating shaft and are supported by mercury lubricated bearings, the shaft rotating at 36,000 rpm. This assembly is called the combined rotating unit. The exhause mercury is then condensed and subcooled by a radiator-condenser and is pumped back to the boiler by means of the high-pressure feed pump. The alternator generates alternating current electrical power at 1800 Hz. A 10-kw unshielded Mercury-Rakine system of this type is expected to weigh about 910 kg (2000 pounds) or 91 kg/kw (200 pounds/kw). It is not expected that space qualified Mercury-Rankine systems will be available before 1970. No specific mission applications have yet been defined.



Reactor-Rankine Cycle Dynamic Power System

RADIOISOTOPE THERMOELECTRIC SYSTEMS

Radioactive isotopes produce heat as a result of the radioactive decay process. In principle, they may be used as a heat source for thermionic, thermoelectric, and dynamic power systems. Thermoelectric systems are described in this topic.

Radioisotope thermoelectric generator (RTG) systems are applicable to long duration power requirements of less than 1 kw where independence from solar illumination or resistance to radiation degradation is a constraint. Specific weights of existing low power (e. g., the 50 watt SNAP 27) systems are in the range of 0. 32 kg/watt (0. 7 lb/watt). Higher power systems are expected to have specific weights of 0. 23 kg/watt (0. 5 lb/watt) or less.[1] Specific costs are strongly dependent upon the cost of the isotope used. The choice depends on the extend of shielding premitted and the operating lifetime. For extended missions an isotope having a long half-life is required in order to minimize the variation in heat output over the variation in heat output over the mission. Characteristics of typical radiosotope fuels are tabluated in Table A.

Since thermoelectric conversion is relatively inefficient (5 to 8 percent) an adequate heat rejection system must be included. The simplest such system consists of fins having a high emissivity coating. Since the heat output of the radiosotope source decays exponentially with time, more heat is produced at the beginning than at the end of the RTG's design life, requiring additional thermal control in order to maintain the hot junction temperature thermoelectric elements at a constant optimum level. The most common technique is the use of a high temperature radiating surface to bypass surplus heat away from the thermocouples at the beginning of the mission. The area of this radiator and hence the amount of heat bypassed is regulated by a thermostatically controlled shutter, the position of which is a function of radiosotope half-life and activity.

A further environmental consideration for RTG power systems is the effect of nuclear radiation produced by the decaying radioisotope fuel on the spacecraft and its payload. (To comply with limitations on the Surveyor Spacecraft the SNAP-11 incorporated shielding weighing 0. 09 kg/watt (0. 2 lb/watt).)

Characteristics of a number of existing RTG power supplies are tabulated tabulated in Table B.[2]

[1] Rappaport, Paul, "Space Power: The Next Step," Space/Aeronautics, 45, Number 4, P. 76, September, 1965.

[2] Barney, R., "Radioisotope Thermoelectric Generators," Research Report No. 14, Hughes Aircraft Company, Space Systems Division, Power Systems Department, September, 1966.

Table A. Typical Radioisotope Fuels

Isotope	Sr-90	Ce-144	Pm-147	Po-210	Pu-238
Half Life (years)	28	0.78	2.7	0.38	89
Power Density (w/cc)	1.1	24.5	1.8	1210	3.9
Source		Fission Products		Neutron Irradiation	
Potential Availability (kwh/yr)	66	800	12	140	4
Lead Time (years)	2-5	1-5	2-5	1-2	2
Estimated Cost ($/thermal watt)	20	1	100	10 to 20	500 to 1000
Shielding Required in uranium Typical Manned System*	(4 inches) 10.15 cm	(6-1/2 inches) 16.5 cm	(1 inch) 2.54 cm	(1 inch) 2.54 cm	(24 inches LiH) 61 cm
Typical Unmanned System**	(0.2 inches) 0.518 cm	(1.5 inches) 3.81 cm	0	0	0

*1 mr/hr at 7.6 cm (3 feet) per kwthermal
**100 r/hr at 7.6 cm (3 feet) per kwthermal

Table B. Characteristics of Existing Radioisotope Thermoelectric Generators

	SNAP 11	SNAP 17A	SNAP 17B	SNAP 19	SNAP 27	SNAP 29
Fuel	Curium-242	Strontium-90	Strontium-90	Plutonium-238	Plutonium-238	Polonium-210
Vendor	Martin-Marietta	Martin-Marietta	General Electric	Martin-Marietta	General Electric	Martin-Marietta
Voltage	28 ±10 percent	28 ±10 percent	28 ±10 percent	24 ±2 percent	29 ±1 percent (voltage regulator)	Not available
Initial power output, watts	25	27.8	26	~50	57 to 56 (end of mission)	400
Weight, kilograms (pounds)	13.9 (30.5)	11.8 (26)	11.7 (25.72)	13.6 (30)	17.5 (38.47) total (includes fuel capsule cask)	~400
Watts/kg (watts/pound)	1.76 (0.8)	2.2 (1.0)	2.2 (1.0)	3.74 (1.7)	2.64 (1.2)	Not available
Kg/watt (pounds/watt)	0.57 (1.25)	2.2 (1.0)	2.2 (1.0)	0.27 (0.6)	0.32 (0.7)	Not available
Efficiency, percent	4.65	5.13, end of life	5.94, end of life	5.1 end of life	4 end of life	Not available
Mission design life	90 days	5 years	5 years	5 years	1-year lunar, preceded by 2-year earth storage	90 days
Hot junction temperature, °F (beginning of life)	1050	1500	1142	842	1100	Not available
Cold junction temperature, °F (beginning of life)	402 (day) 350 (night)	450	320	276	525	Not available
Radiator fin temperature, °F	367	435	310	265	510	Not available
Number of fins	2	6, equally spaced	6, equally spaced	2 (180 degree spacing)	8, equally spaced	Not available
Dimensions, cm (inches)	50.8 (20) diameter x 30.4 (12) long	44.5 (17.5) diameter x 31.8 (12.5) long	4.5 (17.75) diameter x 3.65 (14.38) long; barrel diameter 14.4 (5.67)	56 (22) diameter x 28 (11) long	41 (16.14) diameter x 39 (15.28) long; barrel diameter 13.1 (5.14)	Not available
Watts/ft^3	11.5	16.0	12.5	20.6	31.4	
Remarks		Design and development plan completed. Study terminated.	Design and development plan completed. Study terminated.	Launch late 1967 on Nimbus B	Used for the ALSEP. Fuel capsule inserted after lunar landing.	Initial design phase

RADIOISOTOPE THERMOELECTRIC SYSTEMS

Typical RTG cost as a function of unregulated output power is $5000/watt for limited production and $2200/watt for quantity production.[3]

A 277-watt Strontium 90 powered RTG proposed by General Electric is illustrated in the Figure.[4] Its significant operating characteristics are shown in Table C. The unit weighs 71 kg (156 pounds), distributed as follows:

Heat Source	53.2 kg (117 pounds)
Heat Rejection and Structure	10.4 kg (23 pounds)
Thermoelectric Elements	7.28 kg (16 pounds)
Total	70.9 kg (156 pounds)

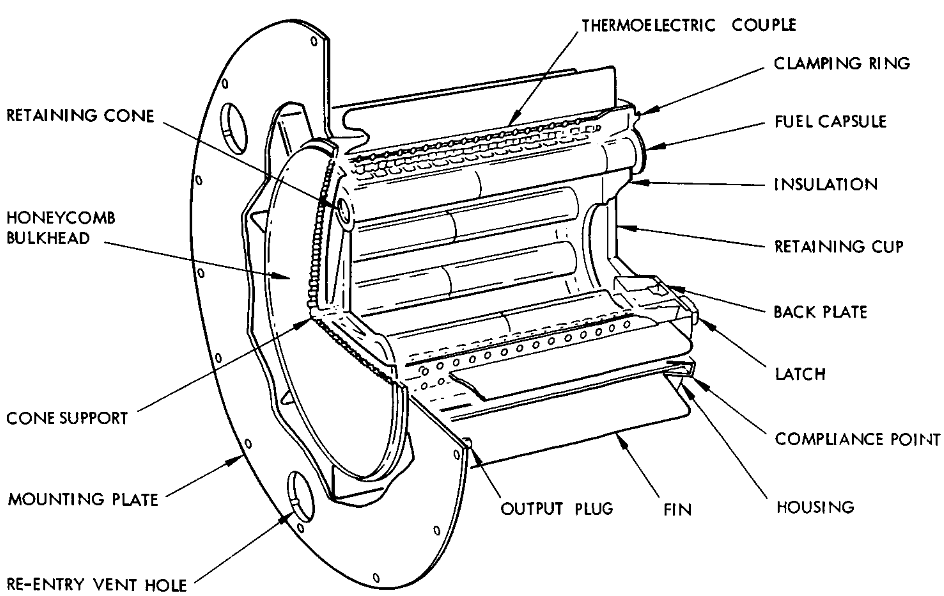
A reliability of 0.99 for one year and 0.95 to 0.75 for 5 years is claimed for this unit.

[3] Harris, E. D., and Dreyfuss, D. J., "Manned Spacecraft Electrical Power Systems: Requirements, Weight Correlation and Cost Implications," Proceedings of the Intersociety Energy Conversion Engineering Conference, Los Angeles, California, September 26-28, 1966.

[4] 250 Watt Radioisotope Thermoelectric Power System, Presentation by the General Electric Corporation, Missile and Space Division.

Table C. 277-Watt Strontium 90 RTG Performance Summary

Operating lifetime	5 years	
Number of thermoelements	320 in 20 modules	
Number of isotope capsules	10	
Reliability		
1 year	0.99	
5 years	0.95 to 0.75	
Electrical-thermal history		
	Beginning of Life	End of Life
Power output (w)	364.0	277.0
Output voltage	28.5	28.5
Hot junction temperature (°F)	1820.0	1700.0
Cold junction temperature (°F)	775.0	740.0
Heat input (w)	7130.0	6317.0
Thermopile efficiency	6.16	5.32
Generator efficiency	5.1	4.38
Average capsule temperature (°F)	2000.0	1890.0
Maximum capsule temperature (°F)	2070.0	1960.0



277-Watt Strontium 90 Fueled RTG

RADIOISOTOPE DYNAMIC SYSTEMS

Radioisotope dynamic systems hold a potential of producing up to 10 kw using a Brayton cycle.

For radioisotope systems developing powers higher than about 1 kw there is a need for higher power conversion efficiency than can be obtained from thermoelectric converters. Because of this need the Brayton gas turbine discussed previously with potential efficiencies as high as 25 percent is the object of great interest. With the Brayton cycle power conversion system, it may be possible to obtain about 10 kw of radioisotope electric power, which is probably an upper limit considering radioisotope cost and availability. The radioisotope Brayton system is of particular interest in future manned missions such as orbital laboratories in which these higher powers are likely to be needed. A projected 11 kw isotope Brayton cycle system weighs 2260 kg (4967 lb) or 204 kg/kw (450 lb/kw) and has an overall efficiency of 21.6 percent.[1] This system is still in the preliminary design stage.

[1] Kirkland, Vern D., and McKhann, George G., "Preliminary Design and Vehicle, Integration of a Pu 238 Radioisotope Brayton Cycle Power System for MORL," Proceedings of the Intersociety Energy Engineering Conference, Los Angeles, California, September 26-68, 1966.

PRIME POWER SYSTEMS

Chemical Power Systems

FUEL CELLS

Fuel cells are efficient power sources for relatively short duration missions.

A fuel cell is an electrochemical device in which the chemical energy of a conventional fuel is converted directly and efficiently into low voltage direct current electrical energy. One of the principal advantages of the fuel cell depends upon the conversion that can (at least in theory) be carried out isothermally, so that the Carnot limit on efficiency of heat engines does not apply. A fuel cell may be visualized as a primary battery in which the fuel and oxidizer are stored externally. The processes are illustrated schematically in Figure A. An actual fuel cell battery of the type developed by GE for the Gemini spacecraft is illustrated in Figure B.[1] This type uses a semi-permeable membrane electrolyte and hydrogen and oxygen as reactants. The hydrogen-oxygen fuel cell has received major emphasis in the manned spacecraft program because of its high efficiency and because it produces potable water as a by-product. Other types of hydrogen-oxygen fuel cells are the Bacon type and the capillary type. Although the differences between these types are basically confined to the cell itself, the operating pressures and temperatures are different, which in turn affects the reactant tank and radiator designs. In the Table power/weight relationships are indicated as a function of mission duration. The volume of fuel cells is 0.84 x 10^4 cm^3/KW day (2.81 ft^3/KW day) for fuel and tankage volume.

As an example of fuel cell cost, a 2 KW system operating for 30 days costs 200/watt.[2]

Fuel Cell Weight per Watt

Days in Operation	Kilograms/watt	Pounds/watt
1	51	113
2	52.8	117
4	86.8	191
5	99.2	218
7	123	271
14	207	556
21	287	632
30	445	980
Fixed Weight	38.6	85

[1] "Fuel Cells—Electrical Power Generation for Space Vehicles," General Electric Corporation, Lynn, Massachusetts, 1963.

[2] Allis Chalmers Manufacturing Company, private communication, May 29, 1967.

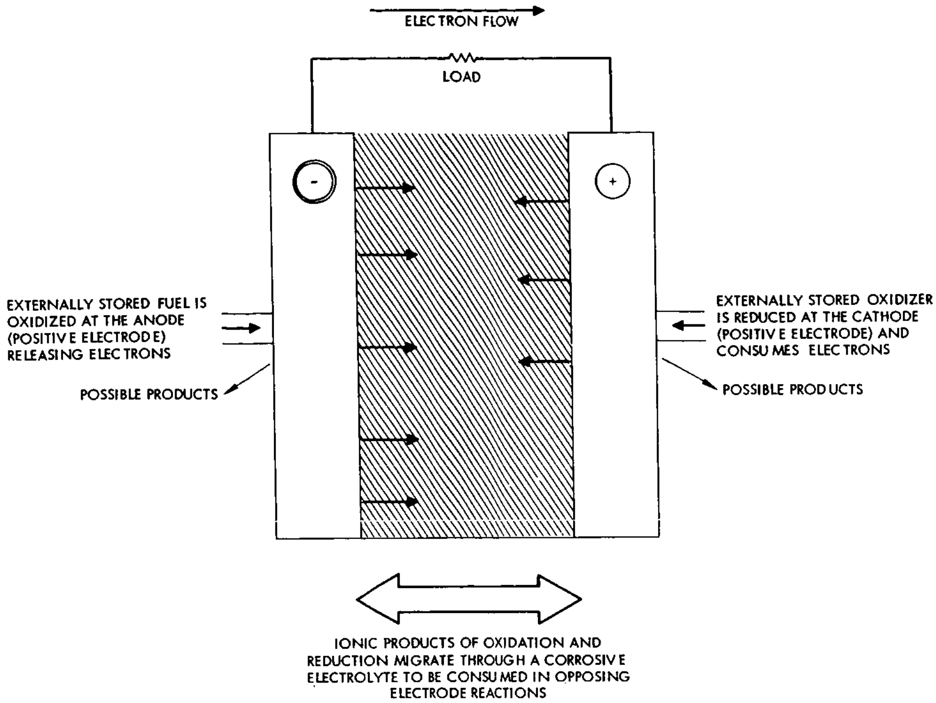


Figure A. Fuel Cell Chemical Processes

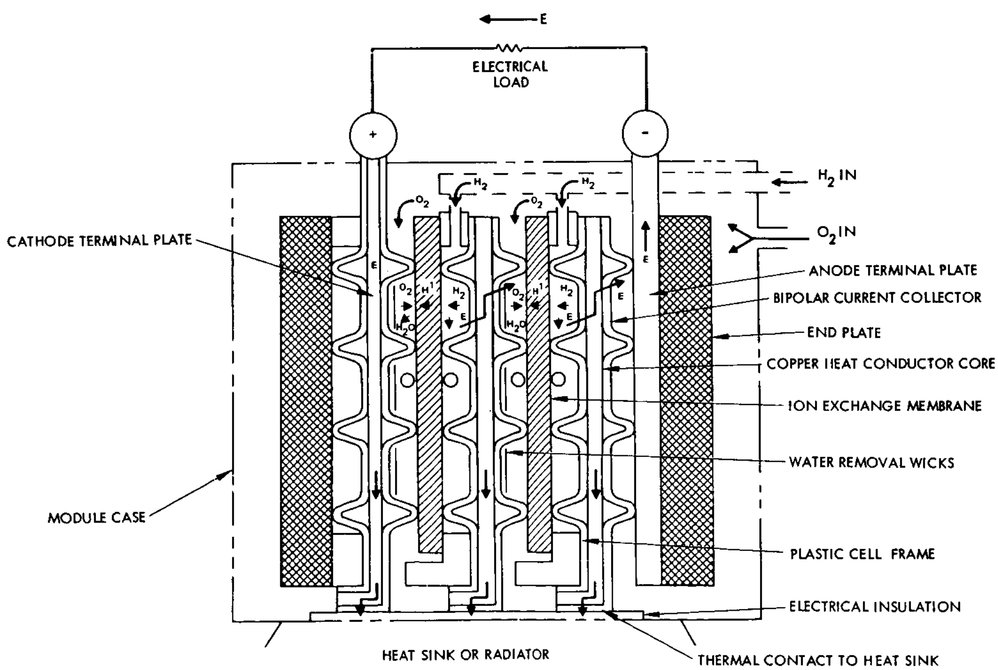


Figure B. Hydrogen-Oxygen Fuel Cell of the Ion-Exchange Membrane Type (General Electric)

BATTERIES

Primary (rechargeable for a few cycles) and secondary (rechargeable for many cycles) batteries are described.

Primary batteries (rechargeable for only a few cycles) are used for spacecraft power when the power levels are low and the mission duration is short. The silver-zinc battery is the most commonly used type, primarily because of its higher energy density (up to 100 watt-hour/lb or 220 watt-hours/kg). However, its loss of charge in storage is severe (40 percent in six months at 25°C). The pertinent characteristics of silver-zinc, silver-cadmium, and other less commonly used space primary batteries are summarized in the Table.[1]

Secondary batteries (rechargeable for many cycles) are required as a part of nearly all space power systems to meet peak power demands and in solar power systems to provide power during periods of solar eclipse. The most useful secondary batteries are the nickel cadmium, the silver cadmium, and the silver zinc types. The nickel cadmium battery has the greatest cycle life but the lowest specific energy. The silver zinc type has the highest specific energy but much lower cycle life. The silver cadmium combines some of the advantages and disadvantages of both these types. The relative performance of these three systems with respect to energy storage density and cycle life are shown in Figures A and B.[2]

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[1] Szego, George C., "Space Power Systems," State of the Art, Institute for Defense Analysis," Washington, D.C., 1963.

[2] Mandel, Hymann, J., Recent Developments in Secondary Batteries," Proceedings of the Intersociety Energy Conversion Engineering Conference, Los Angeles, California, September 26-28, 1966.

Primary Space Batteries

Anode	Zn	Zn	Cd	Cd	Mg	Mg or Ca
Cathode	AgO	Ag_2O_3	AgO	Ag_2O_3	AgCl	Ca CrO_4
Electrolyte	KOH		KOH		KSCN-Ammonia	Fused Salt (Thermal cell)
Separator(s)	Semi-permeable, Nylon + cellophane, Synpor + cellophane, or Cellulosic		Semi-permeable or		Special	Special
Seal or Vent	Automatically activated, with pressure vents, or sealed		Automatically activated and with pressure vents		Automatically activated	Automatically activated
Case	Plastic or Nylon		Plastic or Nylon		Special	Drawn Stainless Steel
Theoretical performance at 25°C:						
voltage/cell	1.82	1.59	1.38	1.16	2.35	
watt-hr/kg (watt-hr/lb)	505 (230)	272 (124)	314 (143)	174 (79)	4.07 (185)	
watt-hr/cm^3 (watt-hr/in^3)	3.72 (61)	1.89 (31)	2.44 (40)	1.34 (22)	2.38 (39)	
Actual performance at 25°C:						
voltage/cell	1.50-1.70	1.2-1.6	1.08-1.30	1.05	2.15	
watt-hr/kg (watt-hr/lb)[1]	110-242 (50-110)	66-198 (30-90)	55-110 (25-50)	44-77 (20-35)	17.6 (8) (-65°F)	
watt-hr/cm^3 (watt-hr/in^3)[1]	0.3-0.6 (5-10)	0.15-0.49 (2.5-8)	0.17-0.37 (2.8-6.0)	0.14-0.26 (2.3-4.2)	0.05 (0.9) (-65°F)	
Wet-stand life, percent loss of charge:[2]					Automatically activated	Automatically activated
at 0°C, 1 month	5	0-5	5	5		
6 months	20	20	20	10		
12 months	50	50	50	20		
at 25°C, 1 month	10	10-15	20	5		
6 months	40	40	50	20		
12 months	--	--	60	35		
at 50°C, 1 week	15	15-20	20	15		
1 month	60	50-60	40	30		
6 months	--	--	--	--		
Cost, \$/watt-hr (at 25°C):					Experimental models	\$500 to \$1000
vented	\$0.35-\$1.00		\$0.60-\$2.00			
scaled	\$0.35-\$1.50		\$0.75-\$2.50			
Available amp-hour sizes	1 to 400 A.H.		0.5 to 300 A.H.			1×10^{-3} to 0.5

1. Dependent upon cell size, discharge rate and number of cycles required. (See Figures 11-34 and 11-35.)
2. Based on percent loss of full and not rated capacity; i.e., calculated as capacity after stand divided by full capacity prior to stand.

BATTERIES

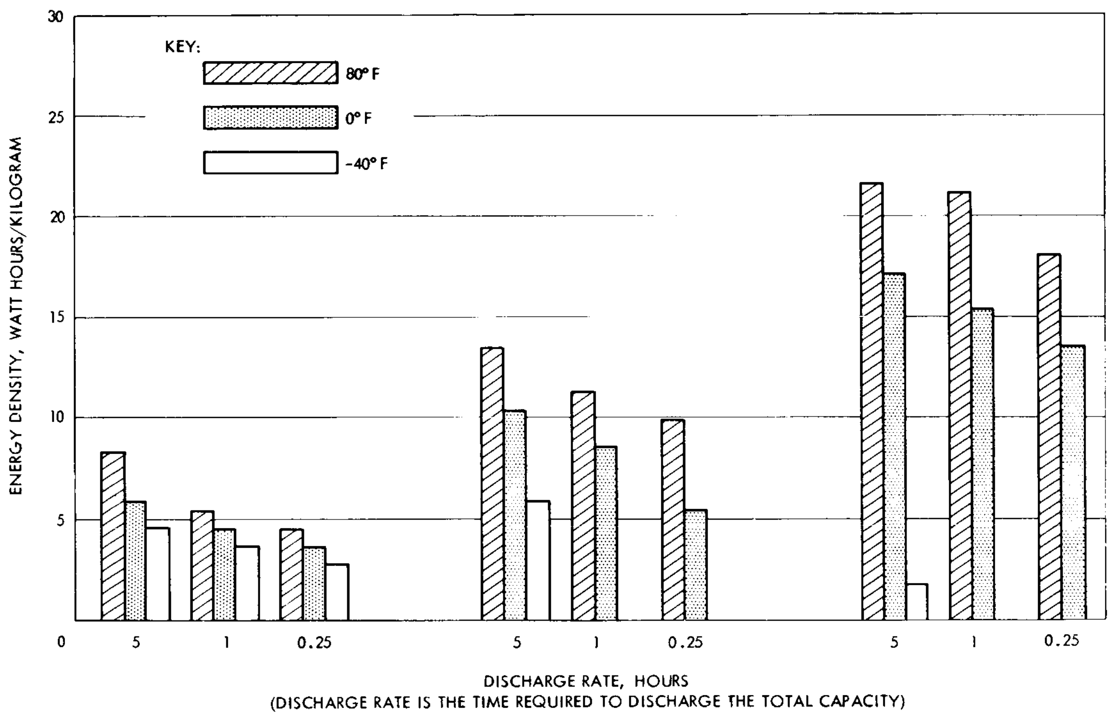

Figure A. Secondary Battery Energy Density versus Discharge Rate

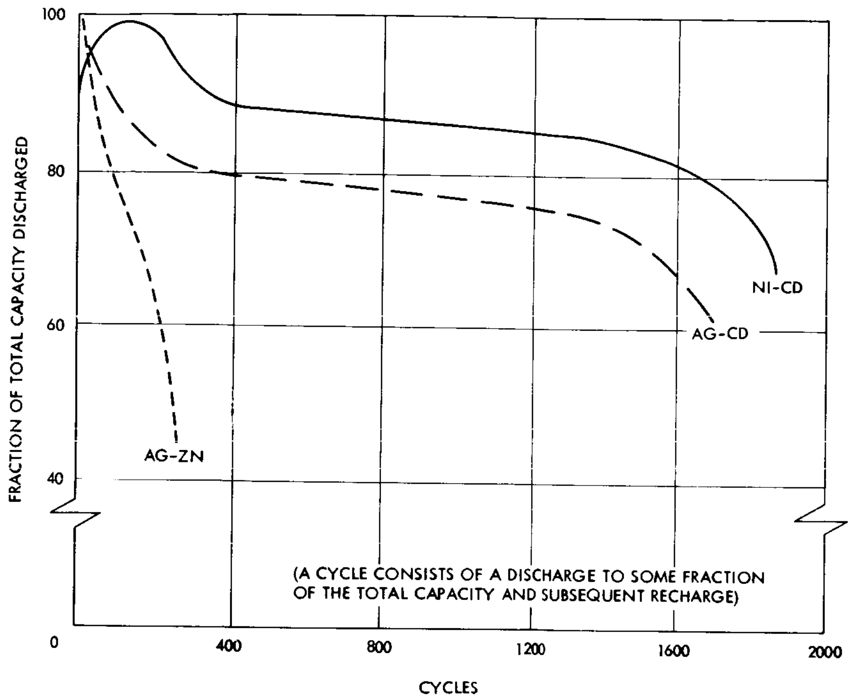

Figure B. Secondary Battery Capacity versus Cycle Life

PRIME POWER SYSTEMS

Power Summary

POWER SUMMARY

COST, VOLUME, AND WEIGHT

Cost, volume and weight relations are given as a function of prime power type, range and power level.

Power summary tables are given for different power levels requirements and different ranges (planets). There are three sets of tables. The first set, consisting of Tables A and B, gives the weight required by various prime power systems measured in kilograms (Table A) and pounds (Table B).

The second set of tables, Tables C and D, gives the volume (or area) of various prime power systems. Table C is the metric system, and Table D presents the same data in English units.

Table E is an estimate of cost for the different system types, ranges and power levels.

Power System Weight, Kilograms

Probe Near:	Mercury				Venus				Earth				Mars			Jupiter		
Distance from Sun, km	57.7×10^6				108×10^6				149×10^6				227×10^6			775×10^6		
Expected Communication Distance from Earth, km	115 to 190×10^6 (1968)				40 to 190×10^6 (1970)				10^4				90 to 310×10^6 (1973)			600 to 890×10^6 (1973)		
Mission Duration, days	82 to 152				72 to 210				30				118 to 260			450 to 1200+		
Power System Type / Output Power, watts	Solar Photovoltaic	Solar Thermionic	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Solar Thermionic	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Fuel Cell	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Reactor Thermoelectric	Radioisotope Thermoelectric
	Total Power System Weight, kilograms																	
10	0.181	0.363		2.27	0.177	0.272		2.25	0.227	4.44		2.25	0.5		2.25	5.8		2.25
25	0.454	0.953		5.67	0.434	0.643		5.67	0.567	11.1		5.67	1.22		5.67	14.4		5.67
50	0.909	1.86		11.3	0.84	1.28		11.3	1.13	22.2		11.3	2.4		11.3	28.9		11.3
100	1.81	3.72		22.7	1.77	2.56		22.7	2.27	44.4		22.7	4.81		22.7	57.8		22.7
250	4.54	9.30		56.7	4.43	6.43		56.7	5.67	111		56.7	12.0		56.7	144		56.7
500	9.09	186.0	311	117.0	8.4	12.8	310	117.0	11.7	222	310	117.0	24.0	310	117.0	289	310	117.0
1000	18.1	37.2	375	227	17.7	25.6	376	227	22.7	444	376	227	48.1	375	227	577	375	227
2000	36.3	74.4	488		35.4	51.3	488		45.4	890	487		96.2	488		1155	488	
5000	90.9	186	1080		84	127	1075		113.2	2220	1080		240	1085		2880	1085	
7500	136.0	279	1625		133	193	1625		170	3330	1625		356	1625		4330	1625	
10000	181.0	372	1665		177	256	1665		227	4440	1665		481	1665		5790	1665	

Notes: 1) Assumes no batteries.
2) Power conditioning losses and weights not included.

COST, VOLUME, AND WEIGHT

Table B. Power System Weight, Pounds

Probe Near:	Mercury				Venus				Earth				Mars			Jupiter		
Distance from Sun, km	57.7×10^6				108×10^6				149×10^6				227×10^6			775×10^6		
Expected Communication Distance from Earth, km	115 to 190×10^6 (1968)				40 to 190×10^6 (1970)				10^4				90 to 310×10^6 (1973)			600 to 890×10^6 (1973)		
Mission Duration, days	82 to 152				72 to 210				30				118 to 260			450 to 1200+		
Power System Type / Output Power, watts	Solar Photovoltaic	Solar Thermionic	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Solar Thermionic	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Fuel Cell	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Reactor Thermoelectric	Radioisotope Thermoelectric
	Total Power System Weight, pounds																	
10	0.4	0.8		5	0.39	0.6		5	0.5	9.8		5	1.1		5	12.8		5
25	1.0	2.1		12.5	0.98	1.42		12.5	1.25	24.5		12.5	2.7		12.5	31.9		12.5
50	2.0	4.1		25	1.85	2.83		25	2.5	49		25	5.3		25	63.8		25
100	4.0	8.2		50	3.90	5.65		50	5.0	98		50	10.6		50	127.5		50
250	10	20.5		125	9.75	14.2		125	12.5	245		125	26.5		125	319		125
500	20	41	685	250	18.5	28.3	685	250	25	490	685	250	53	685	250	637.5	685	250
1000	40	82	829	500	39	56.5	829	500	50	980	829	500	106	829	500	1275	829	500
2000	80	164	1076		78	113	1076		100	1960	1076		212	1076		2550	1076	
5000	200	410	2385		185	283	2385		250	4900	2385		530	2385		6375	2385	
7500	300	615	3578		293	425	3578		375	7350	3578		795	3578		9563	3578	
10000	400	820	3670		390	565	3670		500	9800	3670		1060	3670		12750	3670	

Notes: 1) Assumes no batteries.
2) Power conditioning losses and weights not included.

Table C. Power System Volume, (or Area) cm^3, (cm^2)

Probe Near: / Distance from Sun, km / Expected Communication Distance from Earth, km / Mission Duration, days	Mercury 57.7×10^6 / 115 to 190×10^6 (1968) / 82 to 152				Venus 108×10^6 / 40 to 190×10^6 (1970) / 72 to 210				Earth 149×10^6 / 10^4 / 30				Mars 227×10^6 / 90 to 310×10^6 (1973) / 118 to 260			Jupiter 775×10^6 / 600 to 890×10^6 (1973) / 450 to 1200+		
Power System Type / Output Power, watts	Solar Photovoltaic (Area, cm^2)	Solar Thermionic (Area, cm^2)	Reactor Thermoelectric (Volume, cm^3)	Radioisotope Thermoelectric (Volume, cm^3)	Solar Photovoltaic (Area, cm^2)	Solar Thermionic (Area, cm^2)	Reactor Thermoelectric (Volume, cm^3)	Radioisotope Thermoelectric (Volume, cm^3)	Solar Photovoltaic (Area, cm^2)	Fuel Cell (Volume, cm^3)	Reactor Thermoelectric (Volume, cm^3)	Radioisotope Thermoelectric (Volume, cm^3)	Solar Photovoltaic (Area, cm^2)	Reactor Thermoelectric (Volume, cm^3)	Radioisotope Thermoelectric (Volume, cm^3)	Solar Photovoltaic (Area, cm^2)	Reactor Thermoelectric (Volume, cm^3)	Radioisotope Thermoelectric (Volume, cm^3)
	Total Power System Volume (Cm^3) or Area (Cm^2)																	
10	559			0.858×10^4	0.465×10^3			0.858×10^4	0.65×10^3			0.858×10^4	1.39×10^3		0.858×10^4	16.85×10^4		0.858×10^4
25	1400			2.28×10^4	1.21×10^3			2.28×10^4	1.675×10^3			2.28×10^4	3.53×10^3		2.28×10^4	42.1×10^3		2.28×10^4
50	2700	837		4.56×10^4	2.42×10^3	2.42×10^3		4.56×10^4	3.35×10^3			4.56×10^4	6.98×10^3		4.56×10^4	84.1×10^3		9.56×10^4
100	5400	1580		9.13×10^4	4.75×10^3	4.82×10^3		9.13×10^4	6.63×10^3			9.13×10^4	13.95×10^3		9.13×10^4	168.5×10^3		9.13×10^4
250	13.4×10^3	4000		22.7×10^4	12×10^3	12.1×10^3		22.7×10^4	16.55×10^3			22.7×10^4	34.8×10^3		22.7×10^4	421×10^3		22.7×10^4
500	26.8×10^3	7.9×10^3	45.8×10^4	45.2×10^4	23.9×10^3	24.2×10^3	45.8×10^4	45.2×10^4	33×10^3		45.8×10^4	45.2×10^4	69.7×10^3	45.8×10^4	45.2×10^4	841×10^3	45.8×10^4	45.2×10^4
1000	53.5×10^3	15.7×10^3	92.5×10^4	90.4×10^4	47.6×10^3	48.2×10^3	92.5×10^4	90.4×10^4	66×10^3		92.5×10^4	90.4×10^4	139.5×10^3	92.5×10^4	90.4×10^4	1685×10^3	92.5×10^4	90.4×10^4
2000	107.1×10^3	31.4×10^3	236×10^4		95.6×10^3	96.8×10^3	236×10^4		132×10^3		236×10^4		279×10^3	236×10^4		3370×10^3	236×10^4	
5000	268×10^3	78.7×10^3	790×10^4		23.9×10^3	242×10^3	790×10^4		330×10^3		790×10^4		699×10^4	790×10^4		8420×10^3	790×10^4	
7500	401×10^3	118×10^3	1450×10^4		45.2×10^3	363×10^3	1450×10^4		496×10^3		1450×10^4		1045×10^3	1450×10^4		12600×10^3	1450×10^4	
10000	535×10^3	157×10^3	2340×10^4		477×10^3	482×10^3	2340×10^4		660×10^3		2340×10^4		1395×10^3	2340×10^4		16850×10^3	2340×10^4	

Notes: 1) Assumes no batteries.
2) Power conditioning losses and volumes not included.

COST, VOLUME, AND WEIGHT

Table D. Power System Volume (or Area), feet^3 (or feet^2)

Probe Near: / Distance from Sun, km / Expected Communication Distance from Earth, km / Mission Duration, days	Mercury 57.7 x 10^6 / 115 to 190 x 10^6 (1968) / 82 to 152				Venus 108 x 10^6 / 40 to 190 x 10^6 (1970) / 72 to 210				Earth 149 x 10^6 / 10^4 / 30				Mars 227 x 10^6 / 90 to 310 x 10^6 (1973) / 118 to 260			Jupiter 775 x 10^6 / 600 to 890 x 10^6 (1973) / 450 to 1200+		
Power System Type / Output Power, watts	Solar Photovoltaic (Area, ft^2)	Solar Thermionic (Area, ft^2)	Reactor Thermoelectric (Volume, ft^3)	Radioisotope Thermoelectric (Volume, ft^3)	Solar Photovoltaic (Area, ft^2)	Solar Thermionic (Area, ft^2)	Reactor Thermoelectric (Volume, ft^3)	Radioisotope Thermoelectric (Volume, ft^3)	Solar Photovoltaic (Area, ft^2)	Fuel Cell (Volume, ft^3)	Reactor Thermoelectric (Volume, ft^3)	Radioisotope Thermoelectric (Volume, ft^3)	Solar Photovoltaic (Area, ft^2)	Reactor Thermoelectric (Volume, ft^3)	Radioisotope Thermoelectric (Volume, ft^3)	Solar Photovoltaic (Area, ft^2)	Reactor Thermoelectric (Volume, ft^3)	Radioisotope Thermoelectric (Volume, ft^3)
	Total Power System Volume (ft^3) or Area (ft^2)																	
10	0.6			0.3	0.5			0.3	0.7			0.3	1.5		0.3	18.1		0.3
25	1.5			0.8	1.3			0.8	1.8			0.8	3.8		0.8	45.3		0.8
50	2.9	0.9		1.6	2.6	2.6		1.6	3.6			1.6	7.5		1.6	90.5		1.6
100	5.8	1.7		3.2	5.1	5.2		3.2	7.1			3.2	15.0		3.2	181		3.2
250	14.4	4.3		8.0	12.9	13		8.0	17.8			8.0	37.5		8.0	453		8.0
500	28.8	8.5	16.1	15.9	25.7	26	16.1	15.9	35.5		16.1	15.9	75	16.1	15.9	905	16.1	15.9
1000	57.6	16.9	32.6	31.8	51.3	52	32.6	31.8	71		32.6	31.8	150	32.6	31.8	1810	32.6	31.8
2000	115.2	33.8	83.3		102.6	104	83.3		142		83.3		300	83.3		3620	83.3	
5000	288	84.5	278		257	260	278		355		278		750	278		9050	273	
7500	432	127	510		486	390	510		533		510		1125	510		13575	510	
10000	576	169	824		513	520	824		710		824		1500	824		18100	824	

Notes: 1) Assumes no batteries.
2) Power conditioning losses and volumes not included.

Table E. Power System Cost, Dollars

Probe Near:	Mercury				Venus			
Distance from Sun, km	57.7×10^{6}				108×10^{6}			
Expected Communication Distance from Earth, km	115 to 190 $\times 10^{6}$ (1968)				40 to 190 $\times 10^{6}$ (1970)			
Mission Duration, days	82 to 152				72 to 210			
Power System Type	Solar Photovoltaic	Solar Thermionic	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Solar Thermionic	Reactor Thermoelectric	Radioisotope Thermoelectric
Output Power, watts	Total Power System Cost, dollars							
10	4.30×10^{2}	COSTS NOT AVAILABLE		3.00×10^{4}	3.83×10^{2}	COSTS NOT AVAILABLE		3.00×10^{4}
25	1.08×10^{3}			7.50×10^{4}	9.58×10^{2}			7.50×10^{4}
50	2.15×10^{3}			1.50×10^{5}	1.92×10^{3}			1.50×10^{5}
100	4.30×10^{3}			3.00×10^{5}	3.83×10^{3}			3.00×10^{5}
250	1.08×10^{4}			7.50×10^{5}	9.58×10^{3}			7.50×10^{5}
500	2.15×10^{4}		1.09×10^{6}	1.50×10^{6}	1.92×10^{4}		1.09×10^{6}	1.50×10^{6}
1000	4.30×10^{4}		1.25×10^{6}	3.00×10^{6}	3.83×10^{4}		1.25×10^{6}	3.00×10^{6}
2000	8.6×10^{4}		1.45×10^{6}		7.66×10^{4}		1.45×10^{6}	
5000	2.15×10^{4}		2.00×10^{6}		1.92×10^{5}		2.00×10^{6}	
7500	3.23×10^{5}		2.55×10^{6}		2.78×10^{5}		2.65×10^{6}	
10000	4.30×10^{5}		3.20×10^{6}		3.83×10^{5}		3.20×10^{6}	

Probe Near:	Earth			
Distance from Sun, km	149×10^{6}			
Expected Communication Distance from Earth, km	10^{4}			
Mission Duration, days	30			
Power System Type	Solar Photovoltaic	Fuel Cell	Reactor Thermoelectric	Radioisotope Thermoelectric
Output Power, watts	Total Power System Cost, dollars			
10	5.30×10^{2}			3.00×10^{4}
25	1.33×10^{3}			7.50×10^{4}
50	2.65×10^{3}			1.50×10^{5}
100	5.30×10^{3}			3.00×10^{5}
250	1.33×10^{4}			7.50×10^{5}
500	2.65×10^{4}	5.0×10^{4}	1.09×10^{6}	1.50×10^{6}
1000	5.30×10^{4}	1.0×10^{5}	1.25×10^{6}	3.00×10^{6}
2000	1.06×10^{5}	2.0×10^{5}	1.45×10^{6}	
5000	2.65×10^{5}	5.0×10^{5}	2.00×10^{6}	
7500	3.97×10^{5}	7.5×10^{5}	2.55×10^{6}	
10000	5.30×10^{5}	1.0×10^{6}	3.20×10^{6}	

Probe Near:	Mars			Jupiter		
Distance from Sun, km	227×10^{6}			775×10^{6}		
Expected Communication Distance from Earth, km	90 to 310 $\times 10^{6}$ (1973)			600 to 890 $\times 10^{6}$ (1973)		
Mission Duration, days	118 to 260			450 to 1200+		
Power System Type	Solar Photovoltaic	Reactor Thermoelectric	Radioisotope Thermoelectric	Solar Photovoltaic	Reactor Thermoelectric	Radioisotope Thermoelectric
Output Power, watts	Total Power System Cost, dollars					
10	1.12×10^{3}		3.00×10^{4}	1.35×10^{4}		3.0×10^{4}
25	2.80×10^{3}		7.50×10^{4}	3.4×10^{4}		7.5×10^{4}
50	5.60×10^{3}		1.50×10^{5}	6.75×10^{4}		1.5×10^{5}
100	1.12×10^{4}		3.00×10^{5}	1.35×10^{5}		3.0×10^{5}
250	2.80×10^{4}		7.50×10^{5}	3.4×10^{5}		7.5×10^{5}
500	5.60×10^{4}	1.09×10^{6}	1.50×10^{6}	6.75×10^{5}	1.09×10^{6}	1.5×10^{6}
1000	1.12×10^{5}	1.25×10^{6}	3.00×10^{6}	1.35×10^{6}	1.25×10^{6}	3.0×10^{6}
2000	2.24×10^{5}	1.45×10^{6}		2.70×10^{6}	1.45×10^{6}	
5000	5.60×10^{5}	2.00×10^{6}		6.75×10^{6}	2.0×10^{6}	
7500	8.40×10^{5}	2.55		1.01×10^{7}	2.55×10^{6}	
10000	1.12×10^{6}			6.75×10^{7}	3.20×10^{6}	

Notes: 1) Assumes no batteries.
2) Power conditioning losses and costs not included.

PRIME POWER BURDENS

Power Burdens used for the communications system Methodology are tabulated from the data previously given.

Prime power burdens relate the prime power to weight or cost. They are used in the communications system methodology to determine the lighest or least expensive system. In the communication system modeling, the power supply weight is described by:

$$W_{ST} = K_{W_{ST}} P_{ST} + W_{KE}$$

where:

$K_{W_{ST}}$ = constant relating transmitter power supply weight to power requirement

P_{ST} = transmitter power supply power requirement

W_{KE} = transmitter power supply weight independent of transmitter power requirement

and the fabrication cost is given by:

$$C_{FT} = K_{ST} P_{ST} + C_{KE}$$

where

K_{ST} = constant relating transmitter power supply fabrication cost to power requirement.

P_{ST} = transmitter power supply requirement

C_{KE} = transmitter power supply fabrication cost independent of transmitter power requirement

These constants are summarized in the Table for the power system types discussed in "Prime Power Systems".

Power System Weight and Cost Burden Constants

System	$K_{W_{S_T}}$ kg/watt (lb/watt)	W_{K_E} kg (lb)	K_{S_T} \$/watt	C_{K_E} \$
Solar				
Photo-voltaic	0.0454 to 0.0238 (0.1 to 0.05) (1 AU, 28°C)[1]	Negligible	\$53[2]	Negligible
Thermo-electric	0.0906 (0.2) (1 AU) 0.0498 (0.11) (0.3 AU)	Negligible	*	Negligible
Thermi-onic	0.0454 (0.1) (1 AU) 0.0272 (0.06) (0.3 AU)	*	*	*
Dynamic	*	*	*	*
Reactor				
Thermo-electric	0.0145 (0.32) (20 KW) 0.621 (1.37) (0.5 KW)	181.5 (400)	\$400 (at 5 KW)	$\$1.2 \times 10^6$
Thermi-onic	0.0454 to 0.0227 (0.1 to 0.05)	*	*	*
Dynamic	0.0771 to 0.0113 (0.16 to 0.25)	*	*	*
Radioisotope				
Thermo-electric	0.225 (0.5)	Negligible	\$3,000 (at 1 KW)	Negligible
Thermi-onic	0.0454 (0.1) (above 1 KW)	*	*	*
Dynamic	0.15 (0.33) (Brayton Cycle)	*	*	*
Fuel Cell	0.00386 (0.085) (except fuel)		\$200	

*DEVELOPMENT STAGE: costs or weights not accurately known

1. 1 AU = 1.496×10^8 km

2. Assumes 7 cm x 10 cm, 12 percent efficient cells available at \$10/cell — Helioteks estimate of capability in five years.